THE JOHNS HOPKINS UNIVERSITY
APPLIED PHYSICS LABORATORY
8621 GEORGIA AVENUE
SILVER SPRING, MARYLAND

Operating Under a "Section T" Type of Contract
With the Bureau of Ordnance

STUDIES IN SUPERSONIC AERODYNAMICS AS APPLIED TO CONTROLLED FLIGHT

BY

ROBERT PETER PETERSEN

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Robert Peter Petersen

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A thesis accepted in partial fulfillment of the requirements for the degree of Doctor of Philosophy, Department of Physics, University of California, Berkeley, California.

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ABSTRACT

Measurements have been made on specially streamlined 5-inch rocket missiles in free flight at velocities from 1.85 to 0.9 times the velocity of sound. Ground observation techniques, for supersonic missiles including photo-triangulation, are discussed. Frequency deviation of a high frequency (60 megacycle) carrier has been used for telemetering pressures and accelerations in free flight. Drag measurements using sondes are found to be in good agreement with predictions of drag for simple bodies. The net drag coefficient, referred to body cross section area, $D/q\ S^B$, was 0.81 at Mach No. 1.6 for the streamlined missile. Measurements of base pressure check the von Karman-Moore values for base drag; the coefficient is about 0.20 at Mach No. 1.6. Pressure surveys on GU-2 aerofoils indicate fair agreement with linear flow theory and with the exact theory of Lighthill-Murnaghan at Mach numbers for which the latter theory is valid. These pressure measurements indicate the need for new theory in the transonic regime, Mach No. 1.5 to 1.0, for 10 per cent GU-2 aerofoils. A flip wing assembly using a delay squib has been built which introduces angles of incidence on two wings at pre-selected times of flight. This assembly, by causing angular acceleration, allowed determination of coefficients of lift for GU-2, i.e. double convex, aerofoils. The value of the coefficient, $2.3\alpha/\sqrt{M^2 - 1}$, where α is in radians, is about half that expected from infinite span theory,

$4.2\alpha/\sqrt{M^2 - 1}$. The low value appears to be an effect of finite span. The Schlichting theory which discusses this correction is apparently inadequate. A flip wing assembly was used to cause transverse lift forces; and accelerations and turns have been observed at supersonic speeds.

I. INTRODUCTION

Practical rocket propulsion has been developed to the stage where missiles are now no longer limited to continuous flight at subsonic speeds. Other devices designed to provide more efficient propulsion at supersonic speeds are under study. The present work[1] has provided an affirmative answer to the question of whether aerodynamics would allow controllable flight at supersonic speeds. Measurements have been made on rockets, after they have burned out and during their momentum-sustained flight, in the range of velocities from Mach No. 1.85 to Mach No. 0.7, where Mach number is the ratio of the missile velocity to the local velocity of sound. Aerodynamic lift and drag forces have been measured on these missiles and have been found to be in good agreement with theoretical predictions. Measurements have been made which allow direct comparison with the predictions of supersonic flow theories. Unlike subsonic potential flow, flow in the supersonic regime is characterized by the existence of discontinuities known as shock fronts, and the probable non-existence, for what are normally sufficient boundary conditions, of a uniqueness theorem for the flow. This report is primarily concerned, as is the theory with which we compare our results, with velocities above what is known as the transonic zone.

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1 Because of the war, considerable aerodynamic information possessed by the enemy was not available during the period of this research.

It is, in fact, the supersonic theory,[2] as reviewed and presented by Murnaghan (1)[3] which defines the transonic regime, for it is an essential result of that theory that while no upper limit of velocity restricts its application, there does exist a velocity below which the theory is not valid. This lower limit actually corresponds to the separation of the shock wave from the leading edge of the wing and is principally a function of the required angle of turn of the fluid stream-lines, that is, of the angle of attack which is the angle between the chord line of the aerofoil and the direction of the fluid flow. One experimental method which has been employed gives results that demonstrate some of the characteristics of the breakdown (see Section IV, D). In any case, the transonic region is not necessarily near the velocity of sound, Mach No. 1.0, but may extend up to Mach No. 1.5, 2, or even higher; one may, for that reason, expect some deviation in the present experiments from the predictions of supersonic theories now available.

The nature of these transonic difficulties is not clearly understood. It is well known that aircraft behave very differently as they enter this transonic zone, and to date few pilots

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2 Unfortunately much of the associated theoretical work remains classified and is, therefore, not available to the reader.

3 Numbers in parenthesis identify references in the list given on page 47.

have returned to discuss the experience. Perhaps, therefore, a more reliable basis of observation is the drag of projectiles which demonstrates the unusual nature of sonic speed. A typical curve is presented in Fig. 1.

It is not surprising that sonic speeds should produce unusual effects, since the disturbances created by flight travel through the air at the approximate speed of the missile and tend toward very large discontinuities. For the present, the practical problem of getting missiles into supersonic flight is regarded as a matter of launching through this transonic region at high acceleration, so that, whatever the forces, the impulses on the missile, while in the transonic zone, can be kept small.

The development of supersonic theory of infinite aerofoils is adequately covered by Murnaghan (1)[4]. At the beginning of the present experiments the linear theory of Ackeret (2) was available. No calculations on Busemann's (3) second order approximation were available and Lighthill's paper (4) was in error arithmetically; not until October, 1945, were calculations for infinite aerofoils completed by Edmonson (5).

In practical application, these calculations for infinite span aerofoils are of limited value. However, if we

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4 This reference includes a summary of previous work.

regard the area of a wing forward of the Mach angle, θ, formed at the wing tip (Fig. 2), as essentially undisturbed by the flow from the tip, it is possible to make experiments on the equivalent of an infinite span. The Mach angle, θ, equals $\sin^{-1}(a/V)$. For example, at Mach No. 1.8 θ equals 33.7 degrees, and to explore the infinite span properties along a chord for a wing of chord six inches, the tip must be displaced three and one-half inches from the chord; a span of seven inches, therefore, is needed. For forward parts of the chord smaller spans suffice.

The above mentioned theories may be used to predict the pressures at all points of a given infinite aerofoil. The most detailed and direct test of the theory is, therefore, a point-by-point measurement of the actual pressure developed. By integration of the theoretical pressure over the wing, it is possible to predict the total force which will be exerted by an air stream. Measurement of this total force gives another method of checking the theory. This second method has the advantage that it measures the practical aerodynamic quantity, force on the aerofoil, although it may not reveal the details of the failure in prediction, if disagreement is found.

Three basic flows form the ground work of the present aerofoil theory, namely: Shock waves (6) (6a), normal and oblique, expansion around a corner (6), and compression at a corner (6). Consider the case of a double-wedge wing as

shown in Fig. 3. First an oblique shock wave is constructed at the nose. Such a shock is a discontinuity with the ability to turn a stream through an angle θ, so that the streamlines become parallel to the surface $\overline{AB}$.

It should be noted that at sufficiently high supersonic velocity, the fluid has no "forewarning". For subsonic potential flow, the fluid would already be flowing outward, that is, in effect, it would be forewarned; here, however, the air immediately in front of the corner is absolutely undisturbed while immediately behind the corner it must be moving sidewise because it is proper to regard the wing surfaces as impenetrable. It is, in fact, just this question of forewarning which sets the lower limits on the present theory, because it is true that even at speeds higher than that of sound it is possible for the shock wave to detach and move forward thereby, as it were, warning the fluid of the impending arrival of the object. In the present case, the shock wave is attached to the leading edge of the wing.

The shock theory of Reference 6 shows that the initial fluid conditions P_0, ρ_0, T_0, and the velocity V_0 in the Region 0 on the diagram, combined with the required angle of stream deflection θ, determine the conditions P_1, ρ_1, T_1, and V_1 after shock, in the Region 1 and, therefore, the pressure on the surface $\overline{AB}$. A uniform flow has developed in the Region 1, with known conditions P_1, ρ_1, T_1, and V_1. The flow

turns the corner at point "B" and Prandtl (1) has shown how to derive a solution for such an expansion. It is found that for turning a corner of given angle, a specific set of uniform flow conditions will prevail in the Region 2. The pressure on the wing surface $\overline{BD}$ is thus obtained. Consequently, a complete aerodynamic prediction is available. It should be noted that we have ignored the cross flow at point D and assumed - probably falsely - that in supersonic streams the wing will not "know" what occurs behind it.

To apply the theory to a double convex wing, known as a GU-2 aerofoil from the Italian Guidonia Wind Tunnel tests, it is only required to use a continuous Prandtl expansion, assuming that flow around a curve is essentially flow around successive corners. Murnaghan (1) has demonstrated from the properties of the characteristic lines that this assumption is valid. For all these flows, no uniqueness theorem has been developed as yet and it is not certain that one exists for present theory, thereby emphasizing the radical departure from the incompressible potential flow.

II. DEFINITION OF AERODYNAMIC COEFFICIENTS AND THE EXPERIMENTAL PROGRAM.

While it would be sufficient to obtain the single aerodynamic force acting on an aerofoil or a body, it is convenient to deal with its components parallel and perpendicular to the flow. If the force measured parallel to the flow is

defined as the drag, D, the drag coefficient for the body, C_D^B, is expressed as

$$C_D^B = D/(qS)$$

where q (dynamic pressure) = $\frac{1}{2} \rho V^2$; ρ is the density of the fluid, V the velocity, and S the cross-sectional area of the body. For the wing, the coefficient of drag, C_D^W, is given by

$$C_D^W = D/(qS)$$

where, again, the drag D is the force parallel to the flow, and S is now the planform area of the wing.

If the force measured perpendicular to the flow is defined as the lift, L, the coefficient of lift for a body, C_L^B, may be expressed as

$$C_L^B = L/(qS)$$

where S is the cross-sectional area of the body. The coefficient of lift for the wing, C_L^W, is

$$C_L^W = L/(qS)$$

where S is the planform area of the wing.

In the design of mechanisms for moving the wings and for calculation of over-all dynamic behavior, it is required to know the location of the net aerodynamic forces. This location, known as the center of pressure, is usually expressed in terms of a force moment, M, measured about the leading edge of the wing or, in the case of the body, about the nose. Then, the moment coefficient is given by the equation

$$C_M = M/(qS)$$

where S has the usual meaning, namely, cross-sectional area for body coefficients and planform area for wing coefficients.

The coefficients of the body will depend upon the size, caliber and shape of the missile, on its surface, if the surface gives rise to shock waves, and on the velocity and the properties of the fluid in which motion is taking place. The wing coefficients will depend, in addition to the above variables, on aspect ratio, defined as planform area divided by the square of the mean chord, and on the cross-sectional form. All coefficients will depend on angle of attack, which is the angle between the chord of the wing and the free flow direction. The coefficients of lift are found to be linear with angle of attack, even for angles as large as 10 to 15 degrees.

The first and most important problem to be solved in these experiments was whether any of the predictions derived from the theories already mentioned were valid. For this reason, measurements of the pressure distribution on GU-2 aerofoils were desired to check those calculations. Measurements of lift and drag coefficients for GU-2 aerofoils were also undertaken in order to demonstrate that the predicted forces were actually being obtained on missiles at supersonic speeds. Because of associated work on continuous thrust units and the required balance of thrust and drag, checks were made on the drag of body configurations of simple design. To move wings requires a knowledge of the

moment coefficients; however, measurements on these coefficients have not been included in the present report because of slow development of adequate experimental techniques. Because wind tunnels of supersonic velocities were not available, and also, because scale effects, while probably small, would certainly require some cross-checking with free flight measurements, work on free flight techniques was begun in early 1945.

III. EXPERIMENTAL TECHNIQUES FOR FREE FLIGHT.

A. The Missile

Supersonic speeds could be obtained readily in three ways. Drop test techniques carried out on bombs[5] were available for transonic effects with practical velocities limited to about Mach No. 1.3. Winged missiles could be fired in smooth-bored guns; this method had limitations, however, due to the size and strength requirements of the models. Nevertheless, useful shockwave photographs of such gun-fired models in flight have been obtained at the University of New Mexico[6]. (See Fig. 13). The third method of obtaining supersonic velocity which was the one employed in these tests, was by the use of rockets.

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5 J. F. Bailey of National Advisory Committee of Aeronautics has been conducting such tests.

6 Further details are available in APL/JHU CM Reports— CM-60, CM-69 and CM-129, and in the University of New Mexico Monthly Survey Reports.

The California Institute of Technology developed for the Navy, a 5-inch rocket known as the HVAR, Mk 3, Mod. 3. The powder grain, which has a cruciform shape, weighs 24.4 lbs and has a specific impulse of about 200 lb-sec/lb. The motor which contains the powder weighs 58 lbs. From well-known rocket formulas it is possible to compute the final velocity obtained with any given initial mass of missile. For a missile starting from rest, the maximum velocity attained, V_f, is given to a satisfactory degree of accuracy by

$$V_f = m_p s_i \,/\, (M_m + \tfrac{1}{2} m_p)$$

where m_p is the weight of powder in lbs, s_i the specific impulse in lb-sec/lb, and M_m the mass of missile after burning. This equation is plotted in Fig. 4 along with the curve obtained if a constant drag coefficient of 0.5 is assumed. The correction for drag is seen to be small for the present velocities.

The California Institute of Technology undertook to lighten the motor casing to make possible the addition of wings and control mechanisms, still allowing a sufficiently high final velocity for the experiments, that is, velocities above Mach No. 1.5. Since experimental test conditions, unlike combat conditions, were controllable, especially with regard to the storage temperatures, it was possible to make the new rocket motor 19 lbs lighter. Actually, no accidents have resulted from the use of this new 5-inch HVAR rocket known as the Mod 38. Several photographs of the first type of missile are presented in Figs. 5-a, 5-b, and 5-c, which also show a typical launcher setup. No elaborate launching mechanisms are required,

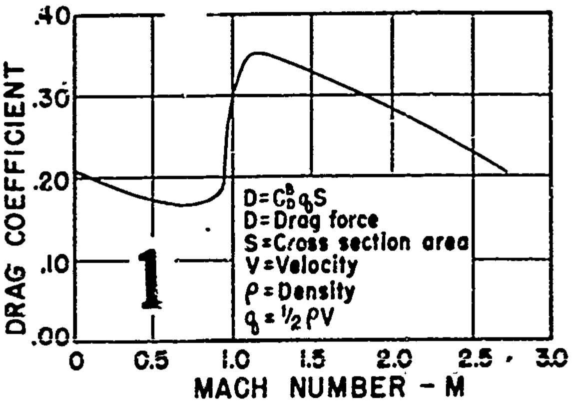

FIGURE 1. DRAG COEFFICIENT, C_D^B, VS MACH'S NUMBER, M, FOR PROJECTILE TYPE 6

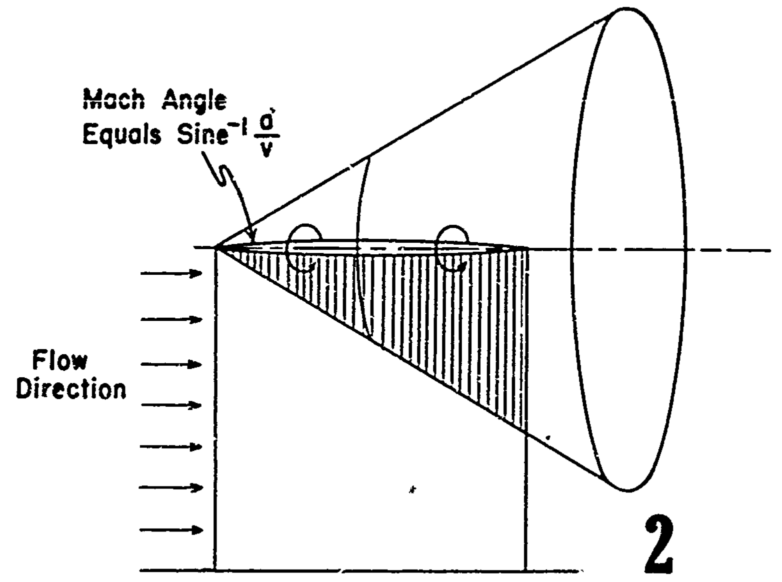

FIGURE 2. THE AEROFOIL AREA IN FRONT OF THE MACH CONE IS NOT INFLUENCED BY THE WING TIP

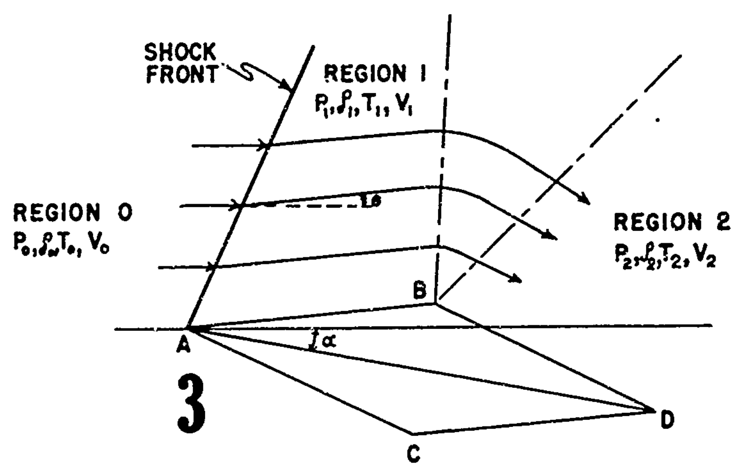

FIGURE 3. THE FLOW FOR AN INFINITE DOUBLE WEDGE AEROFOIL

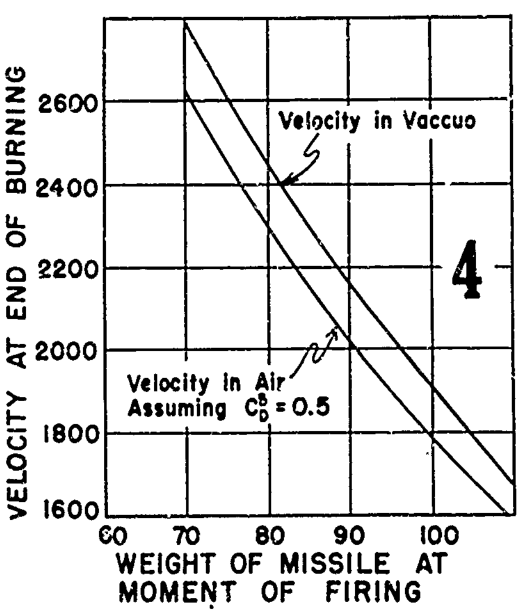

FIGURE 4. MAXIMUM VELOCITY OF AN HVAR 5-INCH ROCKET MISSILE VS INITIAL TOTAL WEIGHT OF THE MISSILE

a pair of iron rails being adequate. At first, wooden beams were used, but their ends tended to burn off, creating unduly large tip-off effects. Tip-off, or change in angle of launching, is caused by the unbalanced torque which occurs as the rocket leaves the ramp when the tail assembly only is supported by the rails, the center of gravity of the missile being forward of the end of the ramp. Ramps of sufficient length to insure high terminal velocity minimize this tip-off effect.

Other rockets and larger test missiles which will become available as work progresses will play an important role in the determination of the aerodynamics of supersonic speeds. They will be propelled throughout flight and their size will allow the addition of more mechanism. However, the 5-inch rockets were available and the present report deals with experiments conducted with them.

In equipping the 5-inch rockets with wings and tail surfaces, close attention must be paid to weight considerations if suitable velocities are to be achieved. For this reason magnesium was used wherever possible. Aerofoils were formed by extruding magnesium through dies and cutting the extrusions to desired lengths. Machining and polishing techniques for this material did not prove to be difficult.

During early firings of 5-inch rocket missiles of the initial design shown in Fig. 5a, it was observed that the whole tail assembly tended to strip off. This tail assembly

was one supplied with the standard Mod. 2, Navy 5-inch rocket, and, in any case, was not a design for which aerodynamic predictions could be made. It was of interest that the fin assembly stripped off during the initial acceleration of 70 g at about 0.7 seconds after firing when the speed was very nearly sonic. The loss of the tail assembly was eliminated by designing a steel sleeve unit as shown in Fig. 6, to which the previously-mentioned extruded aerofoils of GU-2 form, (double convex, and 10 per cent thickness) were attached. A quarter-inch steel shaft brazed to the steel sleeve supplied the primary support with two additional screws and an aligning pin completing the assembly. The sleeve was held to the rocket with Allen set-screws located in two rings as shown. It was found that, in addition to solving the problem of loss of tail assembly, measurements of angle of attack (see Sec. IV D) indicated a uniformly smaller yaw angle in flight which proved important in the pressure survey experiments on aerofoils.

Because of the desire to introduce changes in the angle of attack of the wings and thereby to study the resulting aerodynamic forces, the steel sleeves were also designed to support wings which could be moved during flight. The mechanism for moving these wings is illustrated in Figs. 7a and 7b. It consists of a set of two springs acting on the wings which, together with the aerodynamic lift force, insured that the wings will move up to a mechanical stop. The wings are

5a

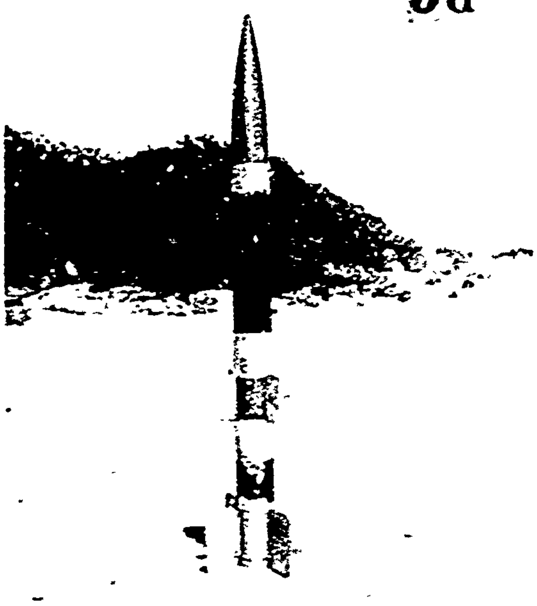

FIGURE 5-a. EARLY TYPE WINGED 5-INCH ROCKET WITH OLD TYPE TAIL ASSEMBLY.

5b

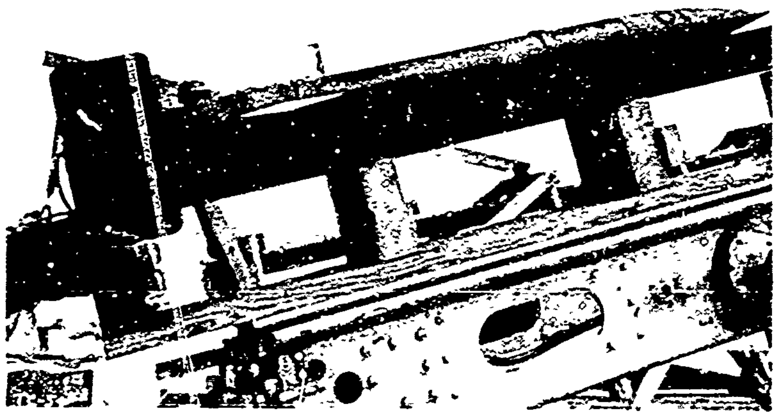

FIGURE 5-b. AN EARLY TYPE WINGED 5-INCH ROCKET MISSILE ON LAUNCHING RAMP.

5c

FIGURE 5-c. A 5-INCH ROCKET MISSILE 0.2 SECONDS AFTER IGNITION; AND WITH 0.8 SECOND OF 5000 LBS. THRUST REMAINING.

held at their initial angle of attack by a piano wire which is led through a cylindrical block containing a piston capable of shearing the wire. The cylinder is driven by means of a small squib semi-cap developed for the proximity fuze. This semi-cap is actually a reduced charge dynamite detonator. It is electrically ignited and can be obtained with different delay times determined by variable powder train lengths. The electrical squib semi-cap is connected to the same firing circuit as the rocket and, at a preselected time, fires, shearing the piano wire, and thereby releasing the wings. Under the action of the spring and aerodynamic forces, the wings move to the desired new angle of attack. In the construction of the sleeve assembly, the alignment of the wings must be held to close tolerances since it is usually desired to avoid high spin. At a velocity of 1700 ft/sec a misalignment of the wing of 1 mil will produce an angular velocity of about 1 rev/sec with the present 5-inch missile.

The original missile whose rough outlines gave rise to spurious shock waves and generally disturbed the flow conditions, has been replaced by the streamlined missile shown in Fig. 8. The smooth cylinder of magnesium tubing which incloses the 5-inch rocket makes clean contact with the steel sleeves previously discussed. The nose is a brass spinning, 14-caliber ogive. The total assembly weighs about 94 lbs at firing. The weight of the rocket jet assembly moves the

center of gravity very far back and so controls the position of the forward wings. This streamlined missile has been used for studies of drag and over-all stability characteristics and, with slight modifications, for pressure survey tests.

B. Data Collection

In the earliest work it was hoped that ground observation by visual means or by radar would be highly effective. The history of such tests, however, should have indicated the really serious difficulties in observing high-speed missiles adequately. Ballisticians have limited themselves to measurements of initial velocity, burst points, and points of fall, using missiles which have small dispersion and high stability due to spin. It was only during the recent war that it has been possible to predict the position of a bullet in space as a function of time of flight. This is a matter quite apart from the point of fall. Indeed, without high speed aircraft and radar, neither the need for such accurate range-time curves, nor their successful use, would have arisen. Reports on the Applied Physics Laboratory's work on fire control have given sufficient clarity to this problem and indicated how inadequate methods of the observation of the projectile flight had been. In the present case the problem was to deal with unstable rocket missiles, having high dispersion, zero initial velocity, and, after one second, a speed comparable to that of a 5-inch projectile after 10 seconds of flight.

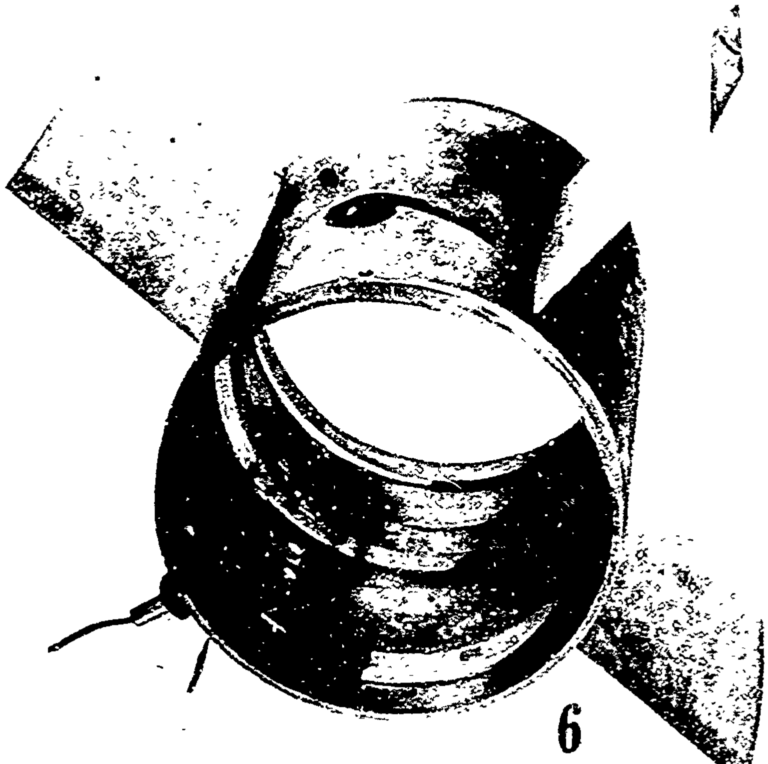

FIGURE 6. STEEL SLEEVE ASSEMBLY.

FIGURE 7-a. SECTION OF A STEEL SLEEVE USED FOR MOUNTING FIXED OR MOVABLE WINGS ON 5-INCH ROCKETS.

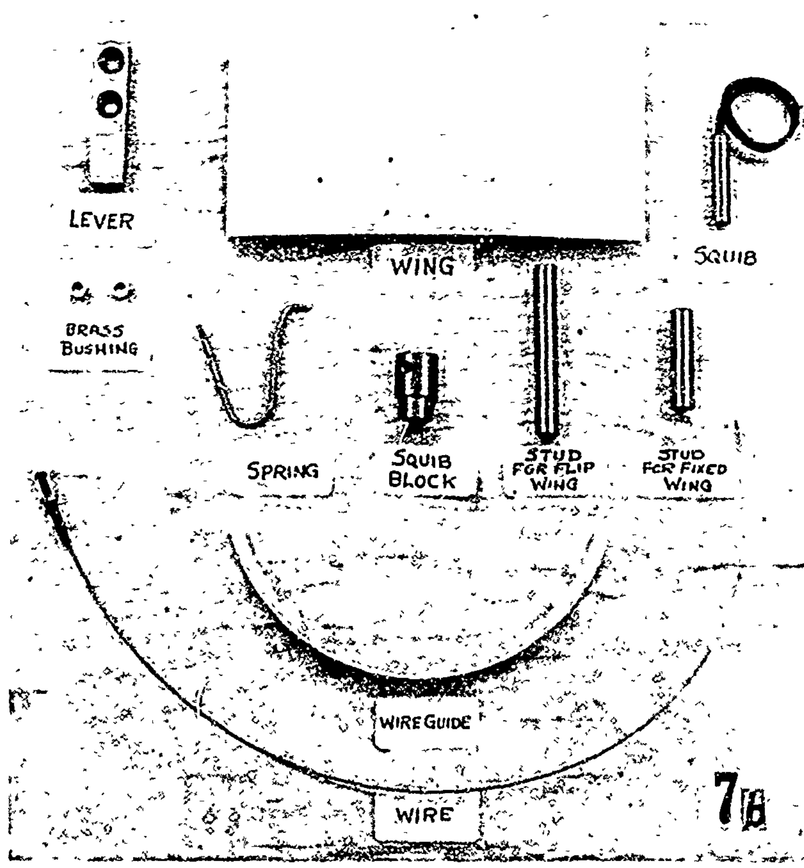

FIGURE 7-b. PARTS USED FOR THE WING FLIPPING ASSEMBLY.

Progress in visual observation techniques is primarily to be credited to the Field Facilities Group under I. B. Irving. This group has successfully tracked rocket missiles from observation stations several thousand yards to the side, the tracking being done using U. S. Navy Mark 51 Director Stands. These stands are moved manually in elevation and train and are equipped with syncros which allow transmission of the angles to conveniently-located dials which are photographed by a camera run by synchronous motors. Tracking cameras are mounted on the Mk. 51 Stands and the error in tracking as determined by these cameras is used to correct the reading of the dial camera. By the use of two of the Mk. 51-Stand tracking stations, it is possible to triangulate the position of the missile. It is estimated that the angular position can be determined to within half a mil. The resulting accuracy of velocity measurement is approximately 5 per cent. Path curvatures obtained from this triangulation method are in agreement with the measures of acceleration which have been made (see Sec. IV F). Drag determinations from Mk. 51-Stand data are not satisfactory except over time intervals of such duration that the velocity intervals become too extensive to be of much value[7].

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7 Further details for this system have been covered in the CM Reports of the Facilities and Propulsion groups of the Applied Physics Laboratory, The Johns Hopkins University.

Another technique which seemed promising was use of the SCR 584 Radar set developed by Massachusetts Institute of Technology and used extensively in the war. Its accuracy and dependability have been disappointing, however. The primary fault, as far as accuracy is concerned, is in the range determination. The fault seems to be related to the band width of the receiver and the method of presenting the information, each influencing the position and sharpness of the radar pip. Ten-yard accuracy from present photographs of the range-scope pip is considered good. The reliability of the SCR-584 radar has been influenced by the terrain and by inexperienced personnel who have had difficulty in picking up the small fast-moving targets which are involved. Because of these general problems very few data from this instrument are included in the present report. Other techniques of ground observation have been tried. Mitchell ribbon-frame cameras were discarded because of the limited extent of the trajectory which they could photograph. Cameras which were motor-driven in angular position were discarded because of the variable time of burning of the rocket and the consequent variation in the space position of the expected initial velocity and because of angular dispersion. Still cameras with rotating shutters were constructed, but were never used because of the necessity of employing night firing[8]. A continuous-wave, Doppler radar of Western Electric

8 The University of New Mexico is conducting tests on drag with this technique.

design was used with good results in the early part of the rocket flights. Its range (about 1500 yards for 5-inch missiles) was so limited, however, that it also has been discarded.

Evaluation of all these methods of ground observation indicated that we could measure path curvatures to an accuracy of 10 per cent, velocity to 5 or 7 per cent, and drag to no better than 10 per cent. Since most of the theoretical results are referred to the dynamic pressure $q = \frac{1}{2} \rho V^2$, it was clear that there existed a fairly serious limitation on accuracy. For this reason, the telemetering techniques which had been under development by most of the aircraft corporations, and by Dr. M. H. Nichols at Princeton University[9], have been employed. (Dr. Nichols was connected with the Section T associated contract at Princeton University).

Frequency modulation of a high-frequency carrier has been employed, the prototype and receiving equipment being obtained from Princeton[9]. (A complete report on this sonde is given in Reference 7). The general arrangement of the equipment is shown in Fig. 9. A Hartley oscillator with a doubler circuit is used. The frequencies of three channels when employed simultaneously are about 55, 63 and 68 megacycles. The end

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9 For a summary of the different methods which might be used successfully the reader is referred to APL/JHU CM-54 by H. E. Tatel.

organs have high impedances since they are used in the high-frequency oscillator circuit. Accelerometer and pressure end organs are available for this system. The time constants of these gauges are satisfactory, the response time of the pressure gauge being several hundredths of a second and of the accelerometer better than 0.1 seconds. Although there is some question about the effects of sidewise accelerations, it is believed that these effects are not serious in the present experiments. The simplicity of this high-frequency, frequency-deviation system, has resulted in better than 80 per cent effective results in tests. Because of the 70-g acceleration, the rugged vacuum tubes developed for the proximity fuze are used in the frequency deviation sonde. The limited life of these tubes and the associated batteries have made necessary careful field techniques to keep the length of time the rocket spends on the launching ramp a minimum. The end organs provide a means of measuring both pressure differences and absolute pressures. Flight times are so short that small leaks and temperature changes during flight have no chance to influence the measurement seriously. Measurements made have been drag, stagnation pressure to obtain velocity, differential pressure to obtain angle of attack, actual gauge pressures to compare with predicted measurements and transverse accelerations to give a measure of lift forces.

The receiving equipment consists of a Hallicrafter S-27

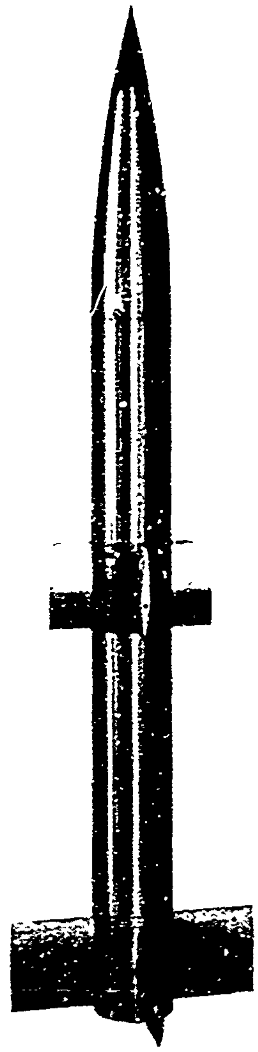

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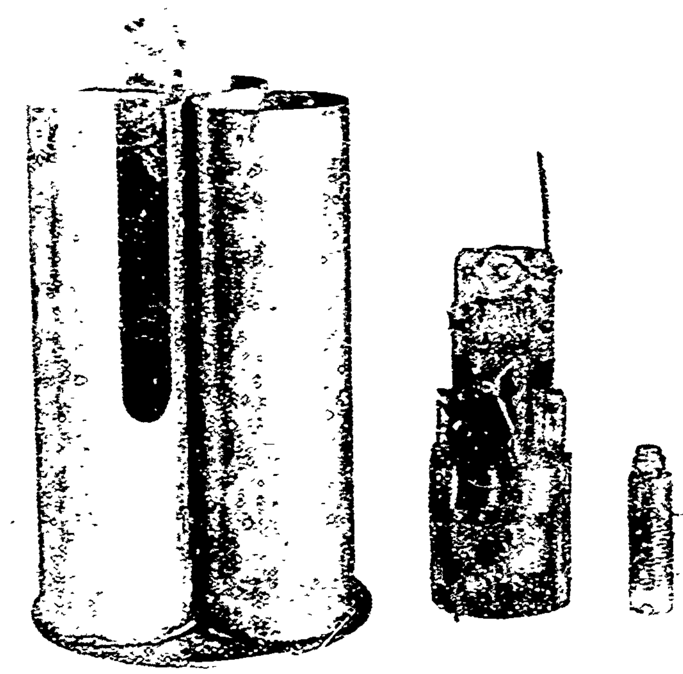
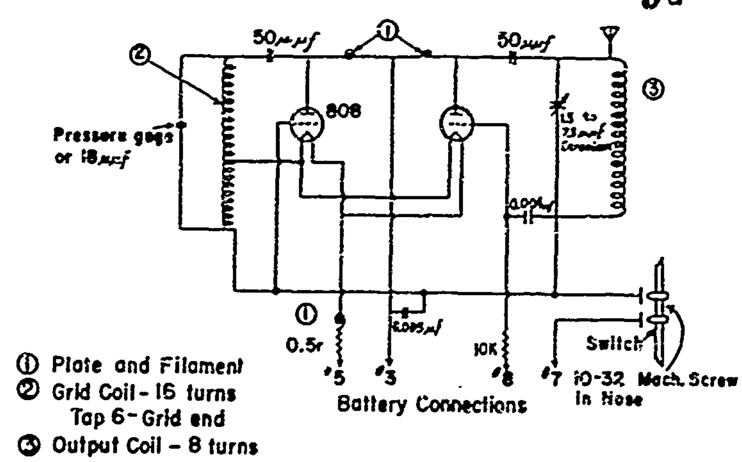
FIGURE 8. THE STREAMLINE 5-INCH ROCKET MISSILE.

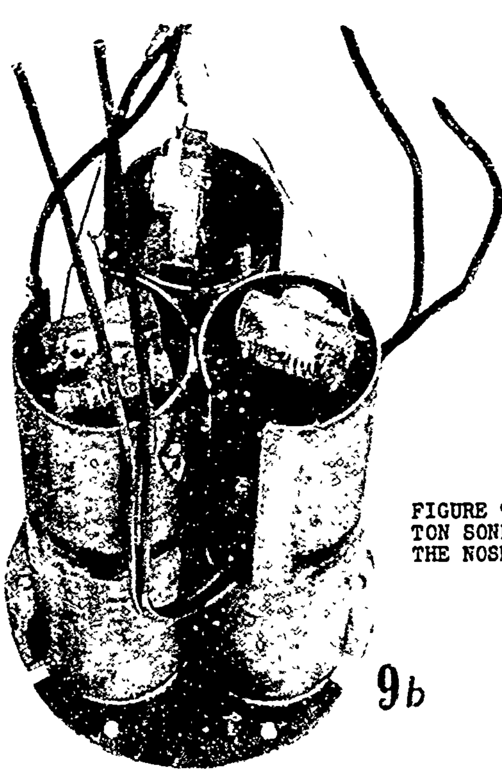


9a



TWO TUBE DOUBLER CIRCUIT

FIGURE 9-a. CIRCUIT AND PHOTOGRAPH OF THE PROTOTYPE SONDE DEVELOPED BY PRINCETON UNIVERSITY.



9b

FIGURE 9-b. AN ASSEMBLY OF 3 PRINCETON SONDES READY FOR INSTALLATION IN THE NOSE OF A MISSILE.

or S-36 with the output of the I. F. being fed into a panoramoscope, which is a commercially available frequency discriminator. The panoramoscope presents the frequency deviation as a displacement of a pip on a cathode-ray scope. A photograph of the experimental setup and of a typical record is included in Fig. 10. Timing marks are added on the side of the film to synchronize the observation with flight characteristics observed at other stations such as the Mk. 51 Stands mentioned previously. Control of the amplitude of the received signals is based on direct observation of the panoramoscope.

In addition, we have added the spin-measurement technique developed by the Applied Physics Laboratory for obtaining ballistic data. The spin sonde consists of a 120-megacycle oscillator feeding a loop antenna which radiates a dipole pattern from the missile. With a fixed dipole, ground antenna, an amplitude modulation is obtained when the projectile spins, the frequency of the modulation being twice the spin frequency. The low cut-off of the receiver amplifier (of the order of 8 cps) required modification of the receiver setup which had previously measured spin of the order of 100 rev/sec but now was desired for low spin rates in the range from zero to 50 rev/sec. Connection was made just aft of the second 686 detector tube, the voltage being fed directly to an oscilloscope deflection plate as well as to a brush recorder used for monitoring. A continuous-film camera was employed. With

this arrangement spin measurements down to 0.5 rev/sec could be made with acceptable accuracy. The use of this spin technique is discussed in the following section.

IV. RESULTS AND DISCUSSION.

A. Lift Measurements for GU-2 Wing

A method of checking the lift-force calculations for double convex (GU-2) aerofoils that has been used, depends on the angular acceleration which two wings produce when they are flipped in such a way as to produce roll.

The general theoretical treatments which have been carried out on aerofoils have been indicated in the first part of this paper. The linear and the Busemann theories predict that the coefficient of lift, L/(qS), of a GU-2 aerofoil is given by

$$C_L^W = 4\alpha / \sqrt{M^2 - 1}$$

where α is the angle of attack measured in radians and M is the Mach number. It is usually convenient to deal with the slope of the lift curve, that is, with $\frac{\partial C_L}{\partial \alpha}$, which is given by

$$\frac{\partial C_L^W}{\partial \alpha} = 4 / \sqrt{M^2 - 1}$$

The more exact calculations by Edmonson (5) give

$$\frac{\partial C_L^W}{\partial \alpha} = 4.3 / \sqrt{M^2 - 1}$$

The values for infinite-span, double-convex aerofoils were first checked in the Guidonia Wind Tunnel where, with the test aerofoil spanning the tunnel, it was found that

$$\frac{\partial C_L^W}{\partial \alpha} = 3.1/\sqrt{M^2 - 1} \text{ at Mach No. 1.85, and}$$

$$\frac{\partial C_L^W}{\partial \alpha} = 3.5/\sqrt{M^2 - 1} \text{ at Mach No. 2.13}$$

In the free flight experiments, missiles were fired with tail fins so adjusted that an angular velocity of several revolutions per second was obtained. At two or three seconds time of flight, the two wings were flipped, as described above, so that the new position of the wings was such as to change the direction of rotation of the missile.

Consider an element of the wing dr located at distance, r, from the axis of the missile, of moment of inertia, I. With a chord of length c, the area of the element is cdr. If is the angle of attack, and S the area of an aerofoil, the lift force acting on the aerofoil is given by

$$L = S\, q\, C_L^W \text{ or as } S\, q \frac{\partial C_L^W}{\partial \alpha} \alpha \text{, so that the lift}$$

force acting on the element of wing is given by

$$dL = c\, dr\, q \frac{\partial C_L^W}{\partial \alpha} \alpha$$

Since this lift force is acting at a distance from the axis of the missile, r, it produces a torque about that axis,

$$dT = c\, q \frac{\partial C_L^W}{\partial \alpha} \quad r\, dr$$

A dynamic angle of attack results from the rotation of the rocket. The angle, α_D, is given by

$$\alpha_D = r\, \omega/V$$

where $r\omega$ is the tangential velocity due to the angular velocity, ω, and V is the velocity of the missile. If the angle of incidence of the wing as measured from the axis of the missile is θ, the angle of attack of the wing will be $(\theta - \alpha_D)$, that is

$$\alpha = \theta - r\omega/V$$

The torque set up by that wing is

$$\Delta T = \int_{r_1}^{r_2} c\, q \frac{\partial C_L^W}{\partial \alpha} (\theta - r\omega/V) r\, dr$$

the integration being made over the wing span. The total torque, which will be the sum of the integrals over all the wings, produces an angular accelerations given by

$$I\dot{\omega} = \sum_{\text{all wings}} \int_{r_1}^{r_2} c\, q \frac{\partial C_L^W}{\partial \alpha} (\theta - r\omega/V) r\, dr$$

If it is now assumed that the lift coefficient is not dependent upon the spanwise position of the section dr, we have

$$I\dot{\omega} = c\, q \frac{\partial C_L^W}{\partial \alpha} \sum_{\text{all wings}} \left\{ \int_{r_1}^{r_2} \theta\, r\, dr - \int_{r_1}^{r_2} r^2 (\omega/V)\, dr \right\}$$

$$= c\, q \frac{\partial C_L^W}{\partial \alpha} \sum_{\text{all wings}} \left\{ \theta \frac{r^2}{2} \Bigg|_{r_1}^{r_2} - \frac{r^3\omega}{3V} \Bigg|_{r_1}^{r_2} \right\}$$

In the present experiments, the values of the various

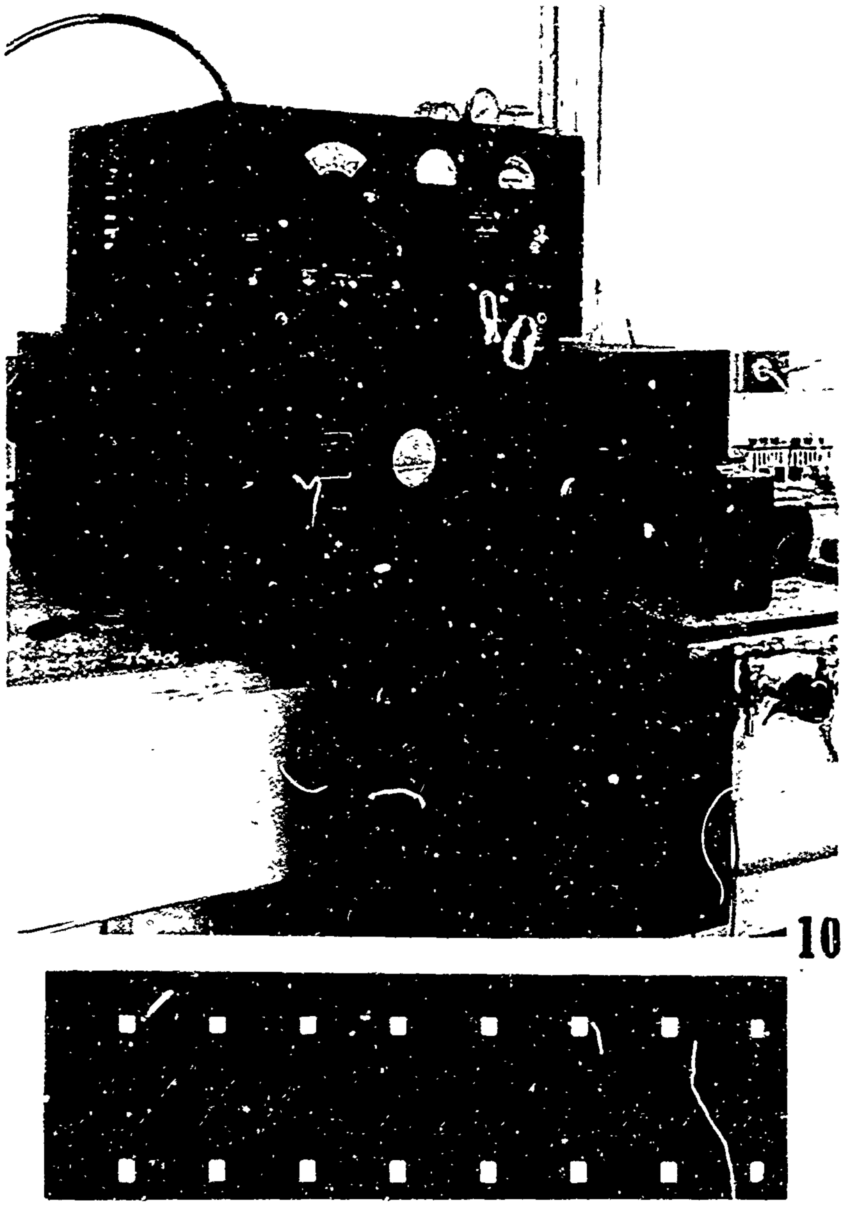


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FIGURE 10. SET UP FOR SONDE RECORDING AND TYPICAL RECORD; HALLICRAFTER, PANSCOPE, AND CAMERA.

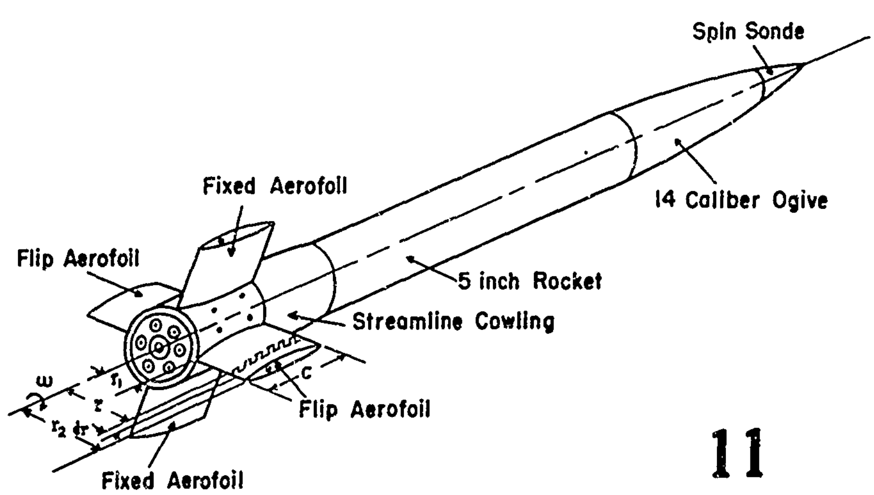

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FIGURE 11. FLIP WING ASSEMBLY IN TAIL OF HVAR.

constants are

$I = 0.0942$ slug ft^2	$c = 0.5$ ft
$\theta = 3^\circ$ for 2 wings	$r_1 = 0.25$ ft
$\theta = 0^\circ$ for 2 wings	$r_2 = 0.50$ ft

The equation of motion for the round becomes (with ω in radians/sec)

$$0.0942\,\dot{\omega} = q\,\frac{\partial C_L^W}{\partial \alpha}\left[4.91 \times 10^{-3} - 7.28 \times 10^{-2}\,(\omega/V)\right]$$

Case (1): As described above, measurements derived from the spin sonde give the angle of roll as a function of time. The slope of the curve gives a curve of angular velocity, ω, against time. The velocity of the round is also measured. A first check on the assumption that $\frac{\partial C_L^W}{\partial \alpha}$ is nearly independent of r is given by checking the above relation at $\dot{\omega} = 0$. From the curves, for Round AH-65, it was found that $\dot{\omega}$ was equal to zero at 1.75 sec, and at this time $\omega = 18.6$ rev/sec and $V \doteq$ 1700 ft/sec. The equation for $\dot{\omega} = 0$,

$$4.91 \times 10^{-3} - 7.28 \times 10^{-2}\,\omega/V = 0,$$

then gives

$$\omega = 19.0 \text{ rev/sec}$$

The agreement suggests that the assumption is valid.

For the spin method of measuring the coefficient of lift a single flip wing assembly is placed at the tail, as shown in Fig. 11. This missile is stable and avoids any interference effects which would be expected if forward wings were used like the missile of Fig. 8. Misalignment errors are also minimized by having the smaller number of wing surfaces.

Case (2): The basic equation can be numerically integrated for different values of the slope of the lift curve, $\frac{\partial C_L^W}{\partial \alpha}$.

One starts the integration with any point on the experimental curve where values of ω_0 and V_0 are measured. From these values, the equation gives $\dot{\omega}_0$. The value of ω at Δt seconds later is given by $\omega_0 + \dot{\omega}_0 \Delta t$. In Fig. 12 typical results of such integrations are given for two values of $\frac{\partial C_L^W}{\partial \alpha}$, along with the experimental curve for Round AH-75. For that round,

$$\frac{\partial C_L^W}{\partial \alpha} = 1.5/\sqrt{M^2 - 1}$$

Other data obtained by this integration method are summarized in the following table:

Round	$\frac{\partial C_L^W}{\partial \alpha}$ for best fit	Mach No. of Measurement	$\sqrt{M^2 - 1} \times \frac{\partial C_L^W}{\partial \alpha}$	$\frac{\partial C_L^W}{\partial \alpha}$ at Mach No. 1.7 in degr
AH-65	2.0	1.60	2.5	.031
AH-75	1.5	1.60	1.9	.024
AH-76	1.3	1.50	1.5	.018
AH-79	1.5	1.50	1.6	.020
AH-129	1.8	1.53	1.7	.022
AH-131	1.8	1.65	2.4	.030

The average value of $\sqrt{M^2 - 1}\ \frac{\partial C_L^W}{\partial \alpha}$ is 1.9 as compared with the theoretical value of 4.3. Since this value is low, consideration was given to the problem produced by the effects of rotation.

Case (3): Due to the dynamic angle of attack resulting from rotation, the aerofoil behaves essentially as though it were warped. That is, the outside part of the wing has an effective

angle of attack $r_2 \omega/V$ while the inner edge has an angle of attack $r_1 \omega/V$. To remove this difficulty, measurements of angular acceleration, $\dot{\omega}$, were made for zero angular velocity. Measurements of $\dot{\omega}$ on Round AH-127 were made which gave an angular acceleration of 25 ± 3 revs/sec^2 at a velocity of 1725 ft/sec and for $\theta = 1$ degree on two wings and 0 degrees on the other two. The value for $\frac{\partial C_L^W}{\partial \alpha}$ then is calculated to be

$$\frac{\partial C_L^W}{\partial \alpha} = (2.9 \pm 0.3) / \sqrt{M^2 - 1}$$

This result is below the theoretical value for infinite span but, as indicated by the estimated accuracy, it is regarded as a fairly reliable determination. We may inquire into a possible explanation of this low value.

In a recent classified publication, Schlichting has suggested the corrections to the infinite span theory which are required for finite span aerofoils. Referring to Fig. 2, Schlichting suggests that in the area behind the Mach angle lift is reduced due to the conical flow originating at the wing tip. By carrying over certain vortex theories from subsonic to supersonic flow, he derives what at present appears to be an empirical relation between aspect ratio and the coefficient of lift for a wing. His relationship is presented in Fig. 13. Supersonic wind tunnel data for infinite span wings from the Guidonia Wind Tunnel, finite span wings from Aberdeen, and data from our free flight test experiments

are plotted on this curve. At present, the only free flight data available are for aspect ratio of 1. Although the wind tunnel tests and the free flight measurements are regarded at this aspect ratio as essentially in agreement, the fact that the Aberdeen data at large aspect ratio goes higher than the theoretically expected value is not understood. The Schlichting theory seems in serious error and further work on the problem is under way to check additional aspect ratios in free flight.

B. Drag Forces on Missiles

As mentioned, work in the laboratory on thrust units required a rough check on supersonic drag calculations. A missile of simple form was chosen for which it was believed drag could be calculated.

The total drag on a missile may be regarded as the sum of the friction, base, and form drags. If these forces are independent one can write, where $C_D = D/qS$, (see page 9) that the net drag coefficient C_D^B is given by

$$C_D^B = C_D^{BW} + C_D^{BS} + C_D^{BB}$$

where C_D^{BW} is the wave drag, C_D^{BS} is the skin friction and C_D^{BB} is the base drag. However, the validity of the assumption that one type of drag is independent of other types of drag is doubtful. For example, the shock wave photographs by the University of New Mexico already referred to (see Fig. 14) show the growth of

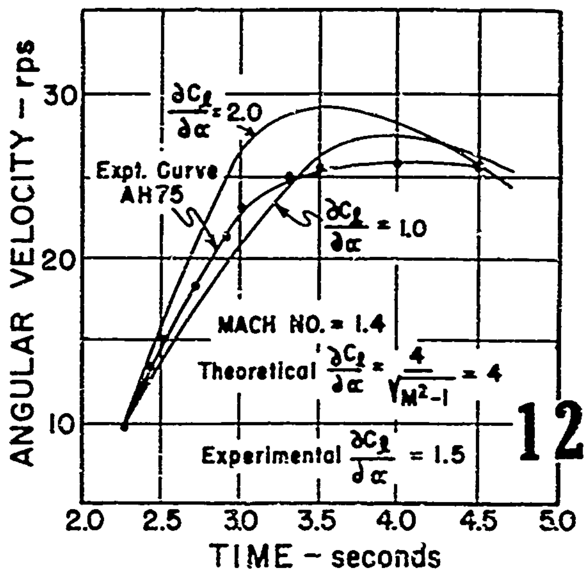

FIGURE 12. ANGULAR VELOCITY OF A 5-INCH ROCKET PRODUCED BY TWO WINGS FLIPPED TO 5° ANGLE OF INCIDENCE

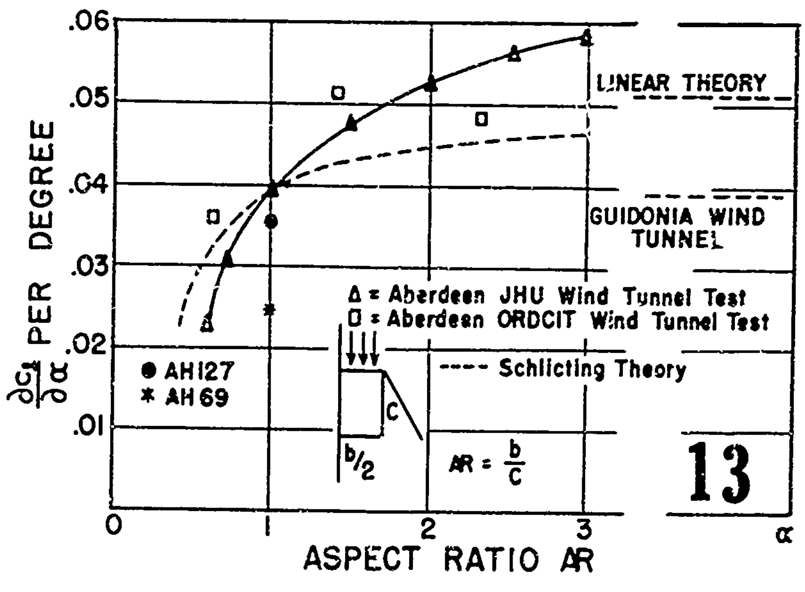

FIGURE 13. LIFT COEFFICIENT SLOPE VS ASPECT RATIO AT MACH NO. 1.7.

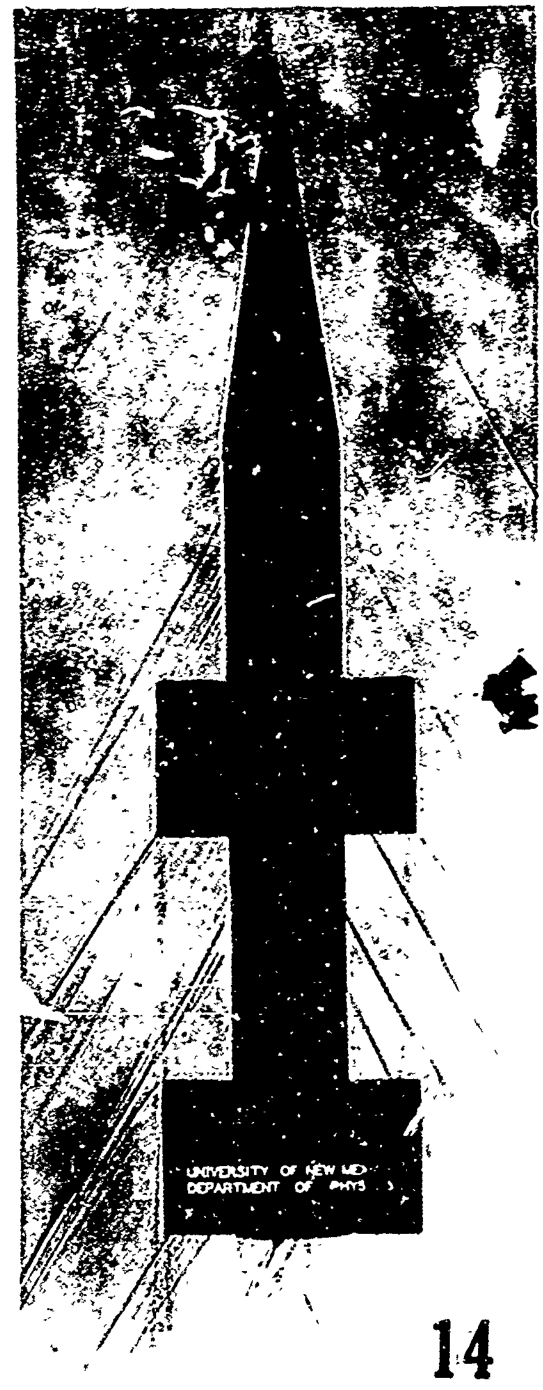

FIGURE 14. SHOCK WAVE PHOTOGRAPH OF A SMALL SCALE MODEL

the boundary layer with increase in the distance, x, from the nose of the missile. Measurements show that the growth is not in disagreement with boundary layer theory which predicts a boundary layer growth proportional to the $\sqrt{x}$. The skin friction is inversely proportional to the boundary thickness and should then be less toward the rear of long missiles. However, these photographs have shown that some severe shock waves are accompanied by a collapse of the boundary layer to a very thin boundary and a subsequent normal growth. Such a thinning of the boundary layer means that wave drag, associated with shocks, and skin friction are not actually independent.

Values for these coefficients of drag have been proposed by previous writers, mostly from work on spinning projectiles. Data on each of the coefficients is discussed in the following sections:

<u>1) Skin Drag:</u>

The friction drag based on Nikaudes formulae for drag on a cylinder is given by

$$D_f = f\,A_s q \quad \text{where } A_s \text{ is the wetted surface, and}$$

$$f = .0008 + \frac{.0552}{R^{0.237}}$$

R, the Reynolds Number, is given by

$$R = \frac{V\,d}{\nu}$$

where V is velocity, d the diameter of the cylinder, and ν the kinematic viscosity, the last-named having the value

$$\nu = 1.58 \times 10^{-4}\ \text{ft}^2/\text{sec}$$

Since in the present experiments, $d = 0.48$ ft and $V = 1700$ ft/sec

$$R = 5.15 \times 10^6 ,$$

and

$$f = .0022$$

For the streamlined missile used (see Fig. 8) the total area of the body and wings is given by

$$A_s = 10.13 \text{ sq ft}$$

Applying the above formula

$$D_f = .0022 \times 10.13 \times q$$
$$= .022\ q$$

then since $C_D^{BS} = D_f/(qS)$

where $S = \frac{\pi(0.48)^2}{4}$

the skin drag coefficient,

$$C_D^{BS} = .121$$

2) Form Drag:

Taylor Maccoll, Von Karman, Tsien, Lighthill and Busemann have all calculated the form drag for zero angle of attack. Tsien has indicated how these calculations may be extended for finite angle of attack, but no numerical evaluations have been made. Expressed in cylindrical coordinates, the supersonic flow theory for a cone at zero angle of attack corresponds to the treatment of the flow for an infinite wedge. A conical shock wave discontinuously deflects the stream into a conical flow, the intensity of the shock being adjusted until the stream

lines of the conical flow become parallel to the surface of the cone. Although a theory for the expansive flow around a corner is available for the two-dimensional case no analysis seems to exist for the expansive flow which occurs at the junction of a conical nose and a straight cylinder. Since determination of drag at a very small angle of attack does not require that the pressure on the straight cylinder be known, the component parallel to the stream in any case being nearly zero, such a calculation is not required for the present experiment.

The pressure which is predicted by the above theories on an ogival nose were first checked by Fowler (8), who employed an ingenious technique. Small holes drilled at the points of interest were connected to the region in which the powder train of a time fuze burned. Since the variation of burning rate with pressure is measurable, the time of bursting of the projectile was a reliable measure of the pressure. Different-sized holes were found to give no large variation above a certain minimum size. Fowler's experiments, some wind tunnel results, and New Mexico shock wave photographs, confirm these calculations at zero angle of attack.

From calculations by L. L. Cronvich (9) the coefficients of form drag for a 14-caliber nose are

$$C_D^{BW} = .119 \text{ at Mach No. } 1.6,$$

and

$$= .139 \text{ at Mach No. } 1.3$$

3) Base Drag:

The action of the high-speed air stream passing the edge of the base of the missile is like that in a diffusion pump. There is created a partial vacuum, the effect of which may be regarded as a drag force because all surface pressures are calculated as gauge pressure. Measurements on base drag coefficients by directly measuring the base pressure are reported in the next section. A partial analysis of this problem has been carried out by Karman-Moore (10) who give as the coefficient of the base drag, and its dependence on Mach number

$$C_D^{BB} = .2\,(1 - .05\ M^2),$$

so that $\quad C_D^{BB} = .175$ at Mach No. 1.6,

and $\quad = .184$ at Mach No. 1.3

4) Drag due to Wings:

Edmonson (5) gives the coefficient of form drag for GU-2 aerofoils at zero angle of attack, based on linear theory calculation as

$$C_D^{W} = .045 \text{ at Mach No. } 1.6$$

$$= .076 \text{ at Mach No. } 1.3.$$

It is to be remembered that C_D^W is referred to the area of the wing.

5) Resulting Total Drag:

The total drag on the missile is then given by

$$D = q\,C_D^W S^W + q\,(C_D^{BB} + C_D^{BW})\,S^B + q\,C_D^{BS} S^B$$

For the streamlined missile the total area of wings, all of GU-2 form, S^W, is 1.5 ft^2, and the cross-sectional area of the body, S^B, is 0.181 ft^2. Then the expected vlaue for the total drag on the missile is

$$D = 505 \text{ lbs at Mach No. } 1.6,$$

and $$D = 428 \text{ lbs at Mach No. } 1.3$$

Since the missile, after burning, weighs 67.6 lbs, the corresponding decelerations expected are 7.5 g and 6.1 g.

6) Experimental Results:

Measurements of the deceleration of a streamlined missile were carried out by decelerometer sondes and Mk 51-Stand triangulation techniques already described. Data is presented in Figs. 15, 16, and 17. In Fig. 15 it is observed that the Mk 51-Stand data can be fitted in several ways and that while the average deceleration over the range is determined within 5 percent, the instantaneous slope as obtained is in considerably greater doubt. In Fig. 16, the sonde record, as taken from flight data, is plotted. The upper threshold of the gauge was at 6.5 g. On the basis of self consistency, and in view of the fact that a mechanical threshold of the accelerometer gauge acts to calibrate the zero point of the sonde in flight, the sonde records are considered accurate to within 5 percent.
In Fig. 17, a log log plot of drag sonde data and Mk 51-Stand data vs velocity is given. The drag sonde indicates that the velocity dependence of the drag coefficient is given by $V^{-0.65}$.

A comparison of the measured deceleration in "g" with the calculations is presented in the following table:

Velocity in ft/sec	Mk 51 Stand Deceleration	Sonde Deceleration	Calculated Deceleration
1700	8.3	7.7	7.5
1300	6.2	6.1	6.1

The agreement indicates that the values of the drag coefficients are known adequately for the conditions of velocity and missile designs used. Further work is under way extending the measurements to missiles flying at angles of attack and also at higher Mach number.

C. Measurement of Base Drag

The pumping action of the fluid stream flowing past the tail creates a partial vacuum on the base of the missile. By direct measurement of the base pressure, it is possible to check the equation of Karman-Moore for base drag coefficient which was considered in the preceding section. It has also been desired that a determination of the effects of a jet on base drag be made and the prove-in of a technique for doing this is a consequence of the present experiments. However, 5-inch rockets burn out so rapidly, and the velocity change with time of burning is so great, that no reliable observation on jet action can be reported from the present tests.

A hole was drilled in that part of the steel rocket casing which extends beyond the jet. A bolt, inserted and

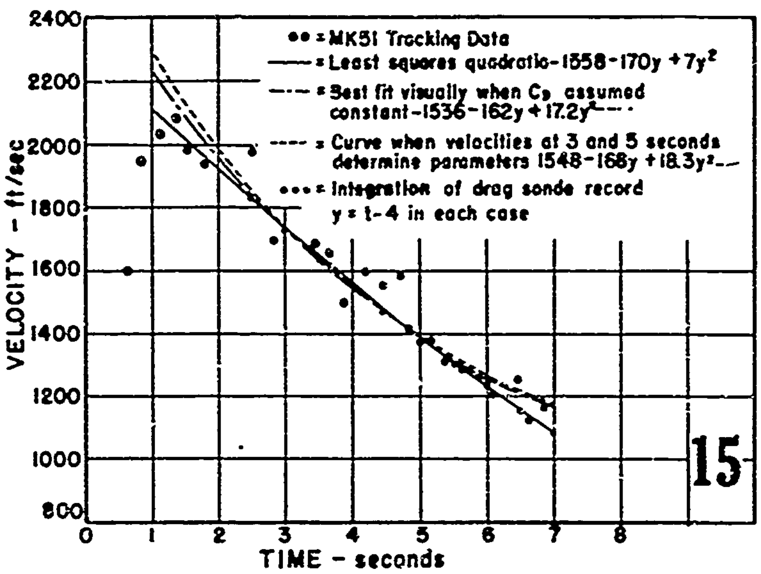

FIGURE 15. MK 51 STAND TRIANGULATION DATA - PLOTTED AS VELOCITY VS TIME.

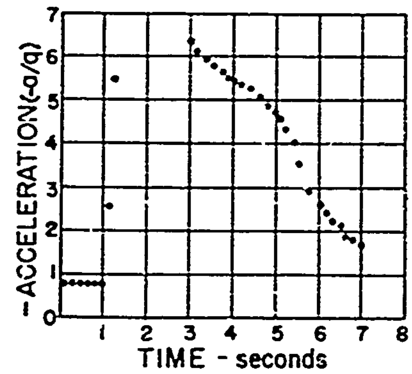

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FIGURE 16. LONGITUDINAL DECELERATION VS TIME FOR STREAMLINE MISSILE.

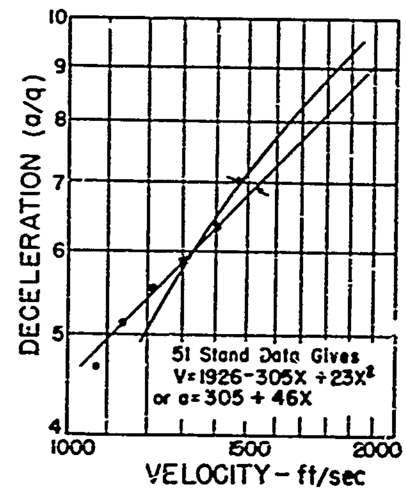

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FIGURE 17. DRAG VS VELOCITY.

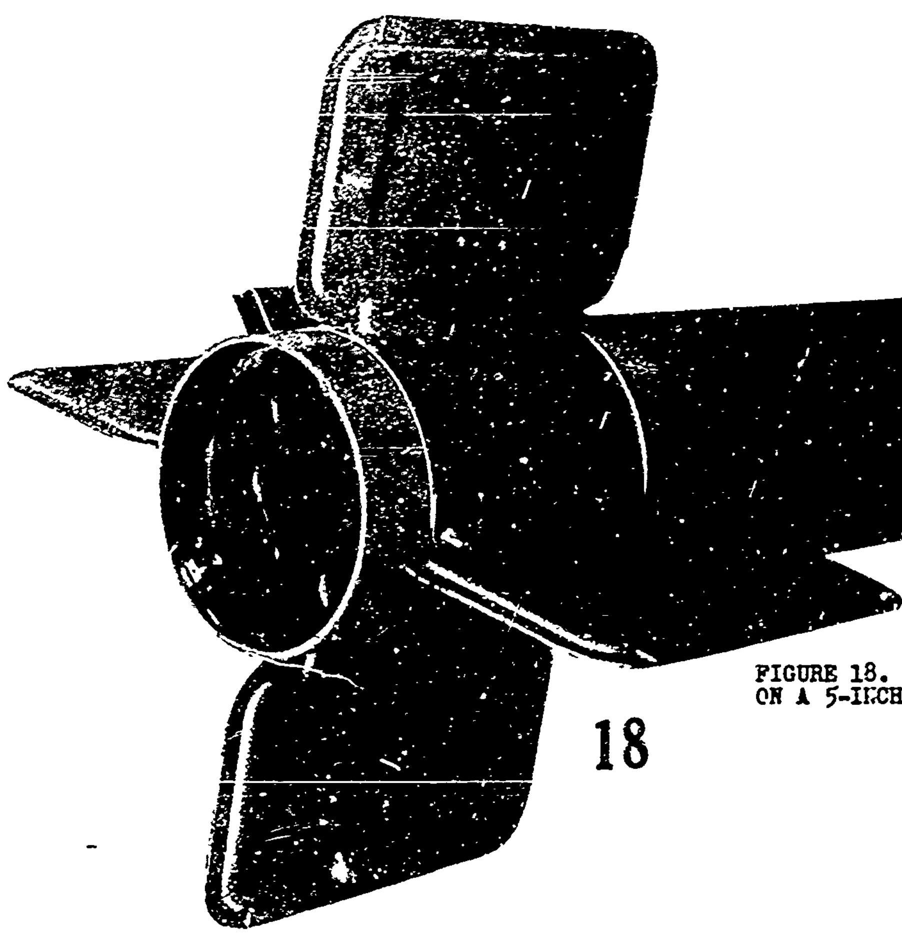


FIGURE 18. BASE PRESSURE MEASUREMENT ON A 5-INCH ROCKET.

tightened in place, had a small hole facing rearward drilled through it. This hole is connected with a copper tube leading to the pressure sonde in the nose. The location of the bolt and hole, and general arrangement of the rocket jets of the missile is shown in Fig. 18. Measurements were made on two rounds and the results are presented in Fig. 19.

The agreement between them is satisfactory and the values obtained are in fair agreement with the equation of Karman-Moore, which is also plotted on the graph. Further theoretical studies are being made to identify the various boundary conditions which are effective in determining the base pressure. Additional tests await the results of such studies.

D. Measurement of Pressure on GU-2 Wing

As indicated in the earlier part of this paper, it appears that the most direct check of supersonic flow theory will be obtained by measurements of pressure. Missiles of the type already referred to have been used for this purpose. A hole of diameter .054 inch was drilled on the midspan line of a 3-inch by 6-inch GU-2 aerofoil at 15 percent or 50 percent or 71 percent of the chord from the leading edge. At the same time a pair of holes located at 27 percent points top and bottom, were connected to a differential pressure gauge to measure angle of attack, or preferably lack of it, since by symmetry arguments alone, zero pressure means zero angle of

attack. These differential pressure sondes on the original rough missile indicated four or five lbs/in^2 which according to linear theory was equivalent to having about four degrees angle of attack. With the new tail assembly, pressure indications are usually less than one-tenth lb/in^2, indicating an angle of attack of the order of one-tenth degree at Mach No. 1.6. A number of rounds have been fired; results are presented in Fig. 20 along with theoretical values for the 15 percent and 71 percent points. The pressures are referred to $\frac{1}{2}\rho V^2$ and plotted against Mach number. The theoretical values plotted are results of computations already referred to in Part I of this report, while the points marked with asterisks come from free-jet tests conducted by Nelson (11) of this laboratory.

It will be noted that rather violent pressure fluctuations occur near the sonic region. In the transonic region, (Mach No. 1.0 to 1.4) the postulates of linear theory hold, but even brutal extension of Murnaghan-Lighthill calculations cannot be made. The fact that, for the 15 percent point, the curves show a change in the nature of the flow as the velocity decreases from Mach number 1.5 tends to verify their development. This lower limit is indicated on the graph. Nelson measured the pressure at ½-inch, 1-inch, and 2 inches from the edge of the aerofoil. The free flight measurement was at 1½-inches. The disagreement observable on the graph is not

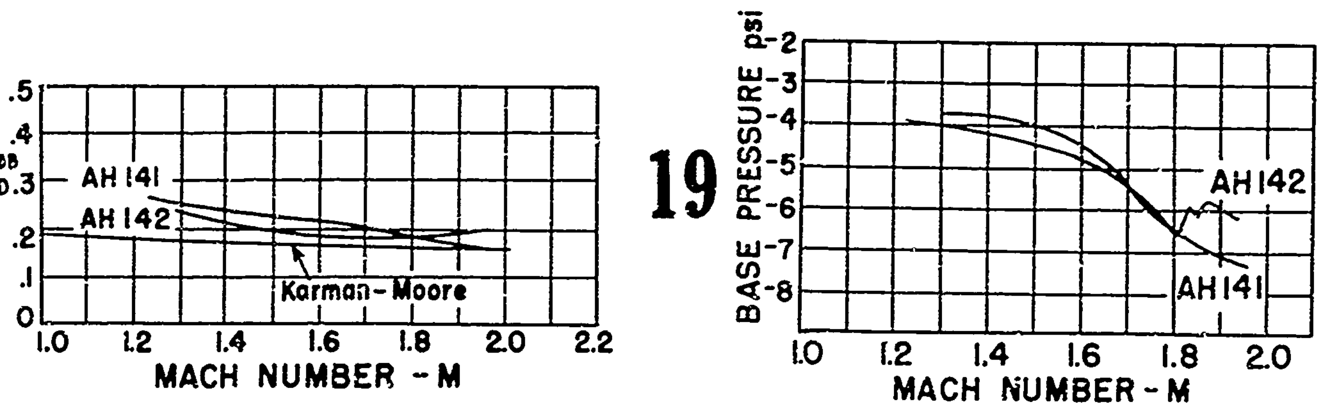


FIGURE 19. BASE PRESSURE AND COEFFICIENT OF BASE DRAG vs MACH NUMBER.

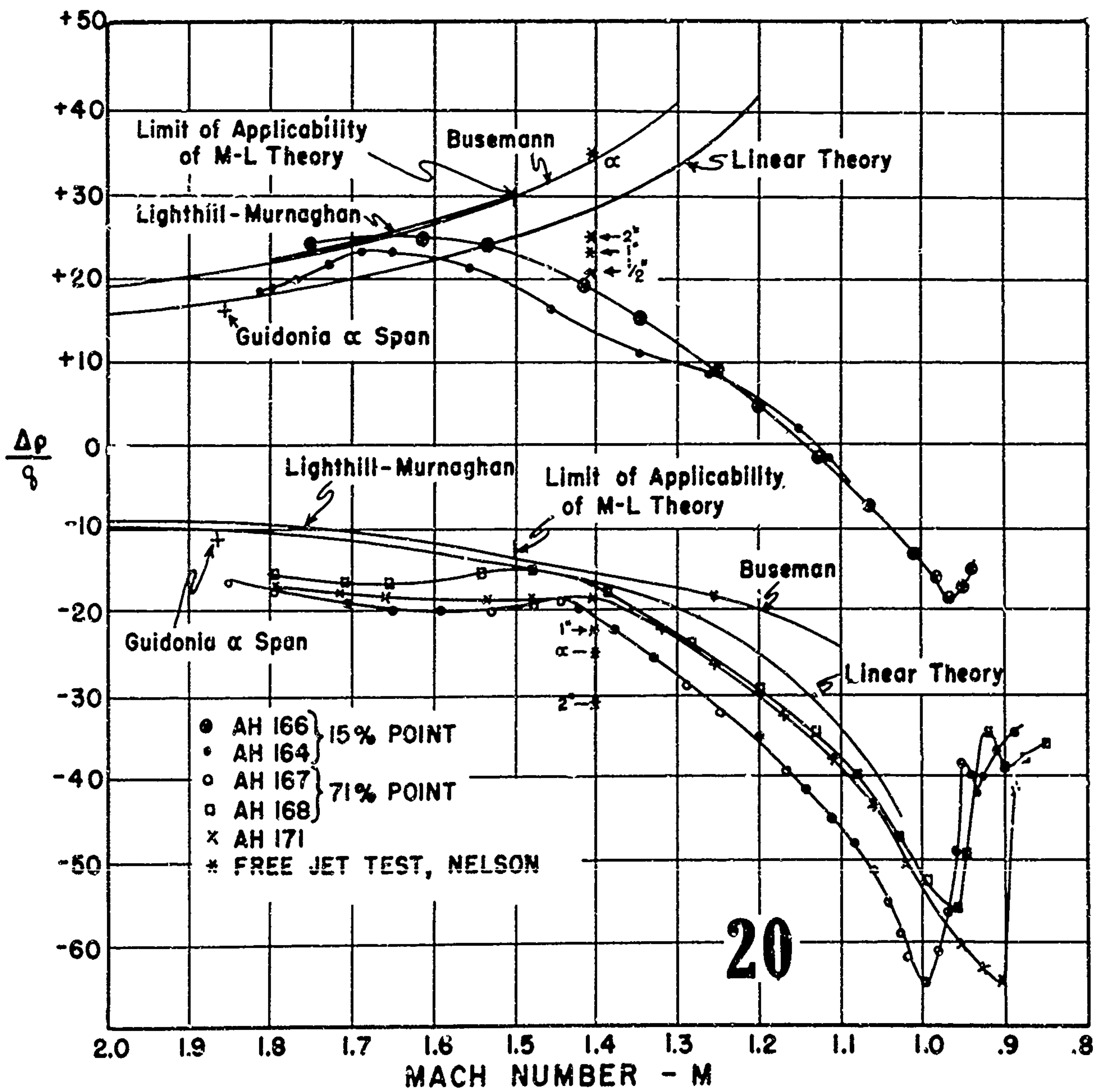


FIGURE 20. THE PRESSURE COEFFICIENT, $\Delta p/q$, VS MACH NUMBER, M, FOR A GU-2, 10%, AIRFOIL

explained, although we feel that the breakdown of the theory is so complete here that it admits of no question of our results.

On the curve for the 71 percent point, the hump at Mach No. 1.47 is real, occurring in all of the three rounds fired. In the region of Mach No. 1.8, the 15 percent point curve is in good agreement with the predicted values; the 71 percent point graph is in disagreement, perhaps due to the finite span effect which at this Mach number can affect this point, but not the 15 percent point. This disagreement may be more serious than appears on casual examination because agreement in the forward part of the wing and disagreement in the rear part means that the hinge moment prediction will be seriously in error.

Further work is under way on wings of greater span to check body effects and clarify the type of breakdown which occurs, but it is believed certain that the basic disagreement indicated by the behavior of the 15 percent point below Mach No. 1.4 can only be answered by new theory.

E. Over-all Missile Behavior

Since an interesting part of studies such as these is their application to complete missiles, it was desired to obtain flight along a curved path. It seemed feasible to do this without elaborate roll stabilization equipment and complicated radio linkage.

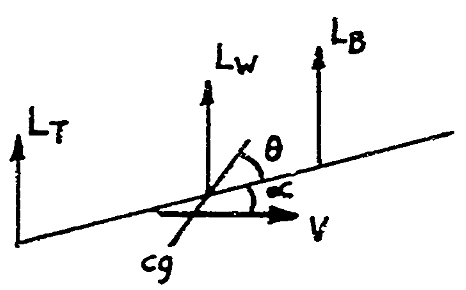
Early attempts at obtaining curved paths by setting the wings at angles of incidence always gave what was visibly reported as a turn, which Mk 51 Stand analysis never confirmed; since the latter is not subject to psychological effects, its indications were taken as correct. (A long background of experience with such work seems to show that observers in this field never become expert.)

Observation of the spin rate of these early flights made it clear that the lift forces were being rotated with regard to the ground observer and that the paths obtained were actually helixes of quite small radius. To offset this effect, it was found that the improved tail fins described on page 14 would hold the spin rate to quite small values (less than 2 rev/sec). Transverse sondes were then installed to measure the acceleration in the direction of lift. This measurement is dependent only on the acceleration due to the lift forces and not on gravity. By introducing angles of attack after several seconds in the manner described above, it is possible to measure the increase in transverse accelerations, and thereby to eliminate residual misalignments which give rise to yaw. Also, the dynamic behavior of the missile is best studied by inserting just such a step function.

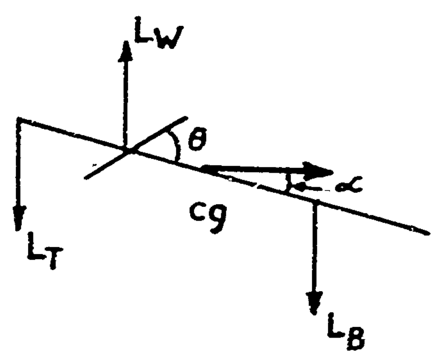
Limiting the study to the static behavior of the missile, for positive angle of incidence of the wings, the missile acquires either a positive or negative angle of attack, depending

on the relative position of the center of pressure and the center of gravity. These cases may be pictured as follows:

Case I



Case II



Here δ is the distance from the cg to the cp of the wings, d_B is the distance from the cg to the cp of the body, and d_T is the distance from the cg to the cp of the tail, we obtain

For Case I $\qquad \delta L_W + d_B L_B - d_T L_T = 0$ as the moment equation,

and $\qquad L_1 = L_W + L_B + L_T$ as the net lift force

For Case II $\qquad \delta L_W + d_B L_B - d_T L_T = 0$

with $\qquad L_2 = L_W - L_B - L_T$ as the net lift,

(The torque due to drag forces is ignored.)

These equations may be applied to a typical round, (Fig. 8) which had the following characteristics:

$\delta = +1.0$ in (Case I)

$d_B = 14.2$ in (taken from Ordcit test at Aberdeen)

$d_T = 20.5$ in

S_T, the area of each tail surface = 0.25 ft^2

S_W, the area of each wing surface = 0.125 ft^2

M, the mass of the missile = 61.6 lbs burnt out

d, the diameter of the missile = 0.5 ft

V, the velocity at squib action = 1750 ft/sec

whereupon q = 3680 lbs/ft^2

For GU-2 aerofoils on wings and tails, we take lift coefficients from Fig. 13 which presents lift coefficients vs aspect ratio at Mach No. 1.7. These are corrected to Mach No. 1.62 by assuming that the theoretical dependence on Mach Number $(M^2 - 1)^{-\frac{1}{2}}$ is correct. Then,

for the wings (AR = 1) $\quad \dfrac{\partial C_L^W}{\partial \alpha} = .041$

for the tails (AR = 2) $\quad \dfrac{\partial C_L^T}{\partial \alpha} = .052$

the Aberdeen Wind Tunnel Tests give

$$\frac{\partial C_L^B}{\partial \alpha} = .0394$$

the equations for lift forces, where α is the angle of attack of the missile and θ is the angle of incidence of the wings with respect to the missile axis, are given by

$$L_W = q\, S^W \frac{\partial C_L^W}{\partial \alpha} (\alpha + \theta)$$

$$L_B = q\, S^B \frac{\partial C_L^B}{\partial \alpha} \alpha$$

$$L_T = q\, S^T \frac{\partial C_L^T}{\partial \alpha} \alpha$$

For this missile, an angle of incidence of 3 degrees resulted from the action of the flip wing mechanism,

so that
$$L_W = 3680\ (.041)\ (.25)\ (3 + \alpha)$$
$$= 34.9\ (3 + \alpha)\ \text{lbs}$$
$$L_T = 95.7\ \alpha\ \text{lbs.}$$
$$L_B = 28.5\ \alpha\ \text{lbs}$$

Solving the moment equation, α is found to be $+0.1$ degree. The lift on the missile then is given by the sum of L_W, L_B, and L_T, which are, for the angle $\alpha = 0.1^\circ$, 108, 9.8, and 3.2 lbs respectively; thus for the total lift, L, = 121 lbs which will produce a lift acceleration of 2.1 g.

The data from the accelerometer sonde appears in Fig. 21, where it is shown that the actual increase is 2.3 g which is in very good agreement with the calculation. The transverse acceleration of 1.0 g already recording probably arises from slight misalignment of nose or tail, which introduces an angle of attack; an angle of 0.35 degrees would produce this acceleration. The sudden but brief rise in transverse acceleration at sonic speed may be of interest.

As a matter of interest, two other sonde records are included in Fig. 22. Because the tail is not a sufficiently simple aerodynamic surface, no adequate comparison with theory is possible. However, these wings were flipped to 5 degrees and were further forward than in previous cases so that greater angles of attack of the missile occurred, producing greater

participation of the tail and body surfaces. The sonde records indicate that a turn of about 5 g would have resulted if roll stability had existed.

F. Stability and an Actual Turn:

Because measurement of the radius of curvature of a turn would be the best check on the measurements of the transverse sondes, an attempt was made to achieve a sufficient degree of stabilization to make such a measurement. A streamlined missile of the type shown in Fig. 21 was built with the non-flipping wings adjustable over small angles. With this small adjustment available, the missiles were placed in a subsonic wind tunnel and trimmed at several hundred miles per hour. Since symmetric aerofoils were being used. no difference in required trim would be anticipated for supersonic speeds. Naturally, after-flow effects might cause a variation, since the behavior of small projections, such as the squib block, could not be evaluated. Several such pre-trimmed missiles were fired with spin sondes and angular velocities of less than 1 revolution in four seconds were obtained. Curved paths were produced, of radius of curvature 16,000 feet, at a velocity of 1600 ft/sec. The acceleration corresponded to 4.9 g which is in close agreement with equivalent sonde records.

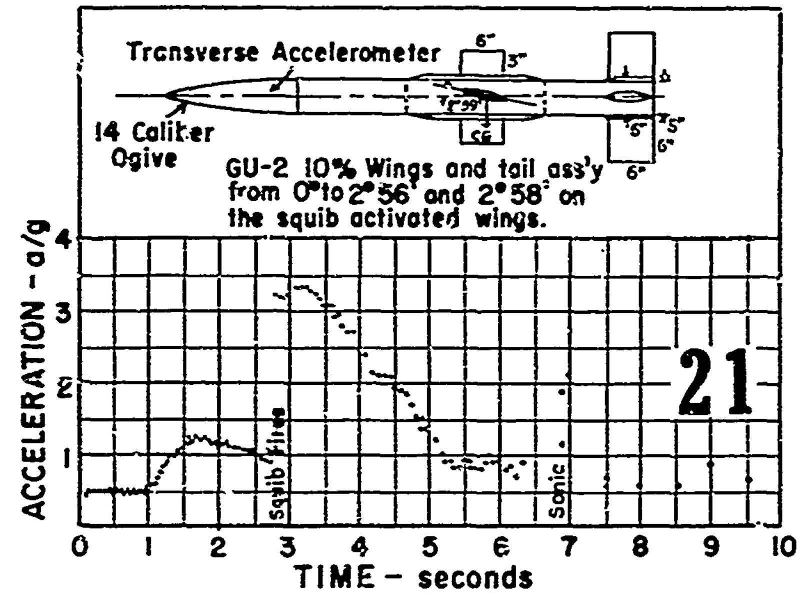


FIGURE 21. TRANSVERSE ACCELERATION SONDE RECORD FOR 3° FLIP WINGS 5-INCH ROCKET MISSILE WITH GOOD TAIL.

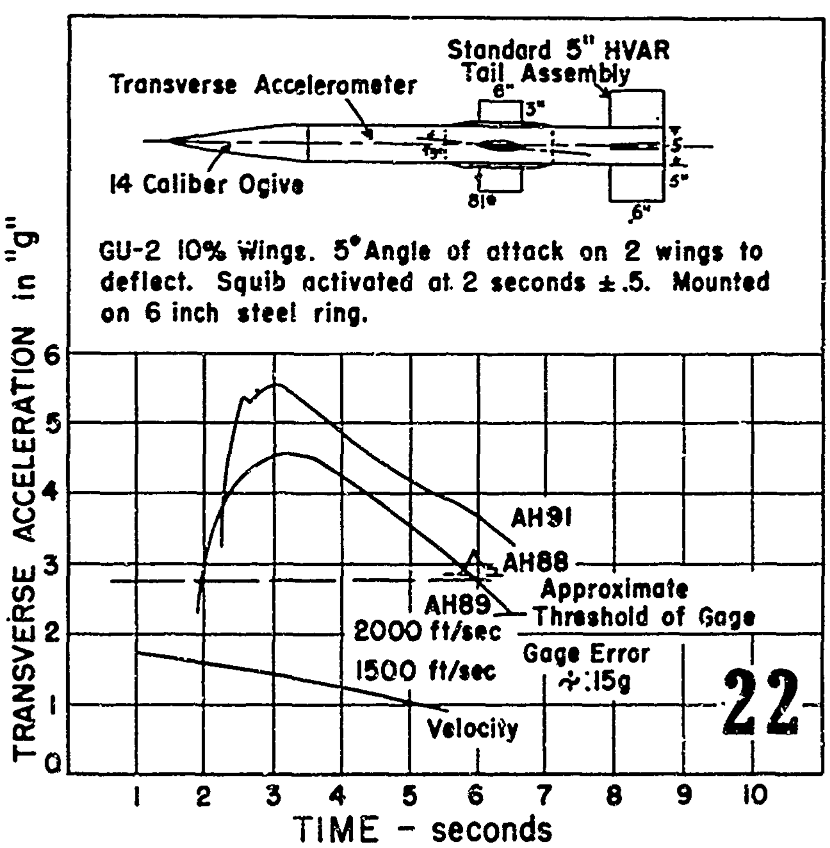


FIGURE 22. TRANSVERSE ACCELEROMETER SONDE RECORD FOR 5° FLIP WING 5-INCH ROCKET MISSILE WITH POOR TAIL AND LOWER GAUGE THRESHOLD SET 2.8 g.

V. CONCLUSION

Results on the aerodynamics of winged 5-inch rocket missiles in free flight may be summarized as follows:

1. The over-all drag force acting on simple bodies calculated by using drag coefficients determined by theory and/or experiment, agrees with decelerometer sonde measurements. The total drag force acting on a streamlined, 5.75-inch, 12-caliber missile weighing 67.6 lbs was found to be 492 lbs as compared with a calculated drag force of 505 lbs at Mach No. 1.6. Specific checks on the base drag coefficient by measurement of base pressure show reasonable agreement with the von Karman-Moore theory. Both indicate a coefficient of about 0.2 at Mach. No. 1.6.

2. A lift coefficient for GU-2 aerofoils has been determined at Mach No. 1.7 and is given by $2.3\alpha/\sqrt{M^2 - 1}$ with α in radians, as compared with infinite span values from linear or exact theory which is about $4.0\alpha/\sqrt{M^2 - 1}$. This lower value is due to the finite span of the aerofoil. Schlichting has suggested a correction for this which seems inadequate.

3. Pressure measurements were made on GU-2 aerofoils to check the predictions of two-dimensional flow theory. The accuracy of the measurements did not distinghish between the values predicted by linear and exact theories. In the region above Mach No. 1.6, these theories and the measured values of pressure are in reasonable agreement

as shown in Fig. 20. However, in the region below Mach No. 1.5 (the transonic region) new theoretical treatments are required.

4. Controlled flight has been achieved to a considerable degree. The angular velocity of roll of missiles at supersonic speed has been controlled by the use of roll flippers. The changes in angular velocities have been determined by spin sonde technique and are in good agreement with theory. Angular accelerations as high as 28 rps^2 have been observed from two 3-inch by 6-inch wings set at 3 degrees. Measurements such as these have been used to determine the lift coefficients of aerofoils. In addition, lift forces of 5 g have been obtained by flipping a pair of wing surfaces. These lift forces have been measured with transverse accelerometers as well as by measurement of the curvature produced in the trajectory.

Further investigations are being directed toward:

1. The location of center of pressure on aerofoils and bodies. This will require more accurate evaluation of the exact vs linear two-dimensional flow theories. It will involve studies of boundary layer and separation phenomena related to the effects of viscosity on flow.

2. A more accurate determination of aerodynamic coefficients with emphasis on base drag of missile with

jets and on three-dimensional flow as it applies on finite span aerofoils and ducts.

3. An investigation of interference effects on aerofoils by forward part of the missile, especially studies of after-flow.

4. Studies of Reynolds Number effects to validate supersonic wind tunnel data or to provide correct scale factors.

5. Development of equipment to increase the accuracy of data, including a long-wave (70 megacycle) continuously-radiating, Doppler shift radar, which will give 0.5 to 1.0 per cent velocity measurement; and a multichannel telemetering system with audio subcarriers capable of use with strain gauges.

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